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SOLAR POWER SYSTEMS FOR SATELLITES IN NEAR-EARTH ORBITS

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ABSTRACT

Since its establishment in 1959, the Goddard Space Flight Center has launched many satellites on various missions. Most of these satellites have used the sun's energy as the primary source of electrical power for the spacecraft. In many cases, the use of silicon solar cells to produce the photovoltaic effect cannot supply sufficient power in the proper time reference, because some orbits contain occulted periods in which solar energy is not available. In other missions, even with continuous solar illumination, it is not practical to supply peak power loads from solar energy alone. Many systems therefore require a capability for storing energy to provide continuous supply of power to the spacecraft. The arrangement which interconnects energy sources, power storage, and spacecraft load forms what is called a solar conversion/energy storage power system.

This paper describes several types of solar conversion/energy storage power systems in use at GSFC: those of the IMP, Nimbus I, and ATS I. Block diagrams show the key features of each system together with its solar configuration.

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INTRODUCTION

In the May 1954 issue of the Journal of Applied Physics a letter to the editor was published concerning a new silicon p-n junction photocell for converting solar radiation into electrical power. Because radiant energy was used without first being converted into heat, the theoretical efficiency was relatively high (22 percent). Devices had been produced with measured efficiencies of 6 percent, in contrast to commercially available photocells with measured efficiencies of 0.5 percent.

In 1958 the National Aeronautics and Space Administration (NASA) was established to explore outer space for peaceful purposes. The eternal scientific quest for the answers to the questions how and why was again to be motivated. Our search for scientific knowledge would no longer be earthbound. We could reach and measure as far out as our technological capability would allow.

The interdependence of these two unrelated events soon became apparent. As the space program developed, NASA's need for a low-weight, low-cost, and reasonably efficient source of electric power was paramount: conventional generating equipment was impractical, energy storage devices were severely limited in their ability to deliver large amounts of power for long periods of time, and the nuclear systems were in their infancy. It was not surprising, therefore, that the new silicon p-n junction technology found an eager customer for its device, the silicon solar cell. Thus, the two developments came together and grew, each supporting the growth of the other.

The solar cell, however, was not a complete solution to the problem of generating electric power in space. True, the sun provided continuous amounts of energy, and the solar cell, through the photovoltaic effect, could convert this energy into electric power with reasonable efficiency. But this was true only if the source was illuminated and if the peak power demand did not exceed the source capability.

Many of NASA's missions required the experiment sensors to operate during periods when the spacecraft would not be illuminated. Others had high peak loads which would require an impractically large number of solar cells. Almost all spacecraft required some form of power conditioning between the source and the loads. The solution to these problems became known as the power system.

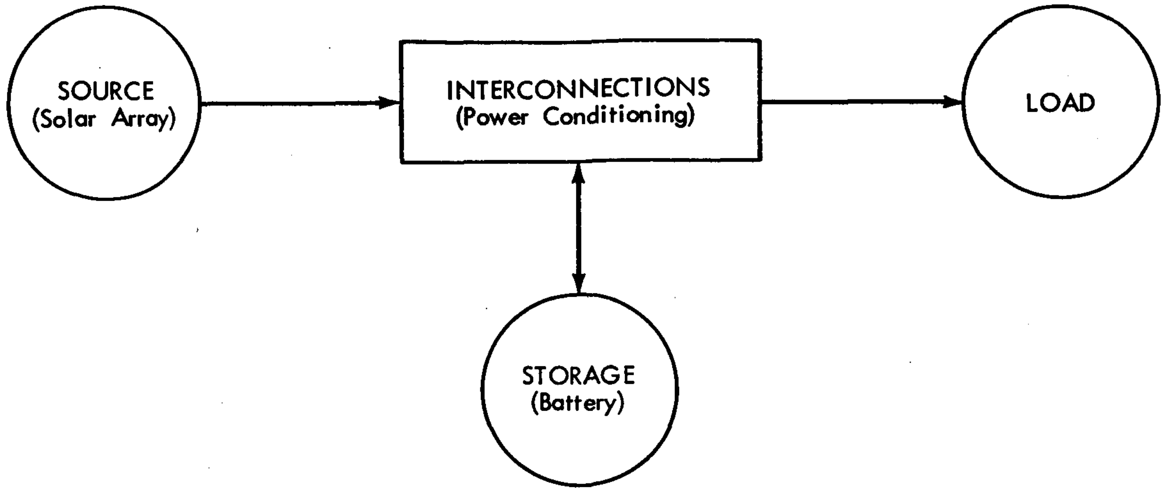


Figure 1. Basic Power System Configuration

The basic configuration of the power system (Figure 1) consists of a source, a reserve or storage element, interconnections, and a load. In most space applications the source has been a matrix connection of solar cells called a solar array. The storage element has been a secondary or rechargeable battery used to supply power during peak loads or during unilluminated portions of the orbit. The interconnections, or power conditioning circuits, contain the regulation, conversion, charge-control, and protective devices necessary to supply continuous power to the spacecraft loads. This type of system is called a solar conversion-energy storage power system.

The system has two basic characteristics:

- The average power capabilities are determined by the size of the solar array and its associated constraints (attitude, orbit, temperature, cell efficiency, and radiation environment).

- The system must operate so as to achieve an energy balance (i.e., energy taken out of the battery must be returned to the battery, with proper allowance for efficiency, if the system is to operate for long periods).

These characteristics are so basic they are often forgotten. The second is especially critical to spacecraft systems requiring active attitude control to enhance power capability (for example, Nimbus, the Orbiting Geophysical Observatories, and the Orbiting Astronomical Observatories).

This paper discusses three of the solar conversion-energy storage power systems flown by the Goddard Space Flight Center: those on Explorer XXXIII, Nimbus II, and the Applications Technology Spacecraft, ATS I. Although the basic concepts of the systems are identical, the implementation of these concepts varies with such parameters as attitude, stabilization, and mission. Each system is discussed with respect to mission objectives, solar array, battery, power control and distribution, and performance.

EXPLORER XXXIII

Explorer XXXIII, also known as the Anchored Interplanetary Monitoring Platform (AIMP), is typical of a class of small interplanetary monitoring platforms (IMP) used in the scientific exploration of solar-terrestrial relationships (Figure 2). Project IMP consists of a series of seven spacecraft designed to study the radiation environment of the interplanetary medium, the interplanetary magnetic field, and the magnetic field's relationships with particles from the sun in the vicinity of the earth and the moon.

The AIMP portion of the series is designed to investigate and monitor the radiation environment of the interplanetary medium in the vicinity of the moon, and to measure interplanetary and lunar magnetic fields. The intent is to anchor an IMP type of spacecraft in a lunar orbit to enhance its effectiveness in

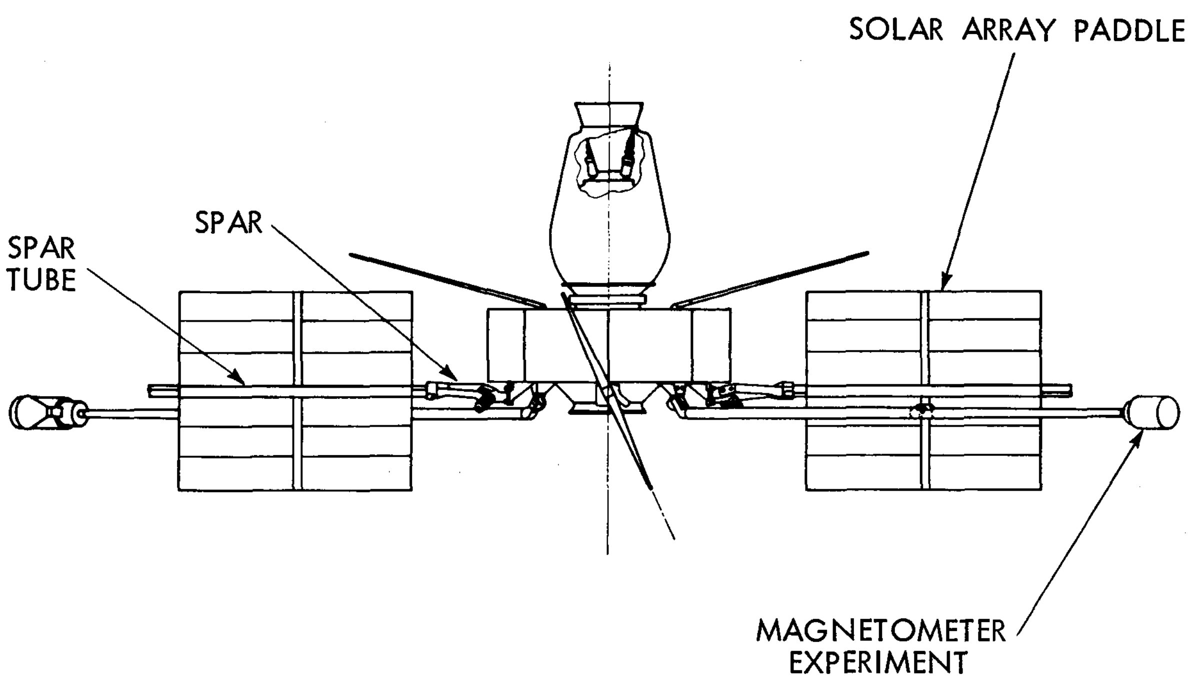


Figure 2. Explorer XXXIII (AIMP)

collecting data. This lunar orbit permits data coverage of the sun-side and shadow-side of both the moon and the earth every 29.5 days. This orbit compares favorably with that of a high-apogee eccentric earth orbit which requires 1 year to get the same coverage.

Power System Description

The power system for the AIMP spacecraft consists of the following major hardware components: four solar array paddles, a storage battery, charge control, and regulation equipment. The system was designed to provide at least 35 watts of continuous average power for not less than one year. Figure 3 shows the block diagram for the interconnection of these components. Table 1 lists the weights of the various components of the system. Since attitude control of the spacecraft is maintained by spin stabilization, no major amount of power is required to maintain spacecraft attitude. Accordingly one of the key features of the system is the ability to turn off about 95 percent of the power without loss of the spacecraft.

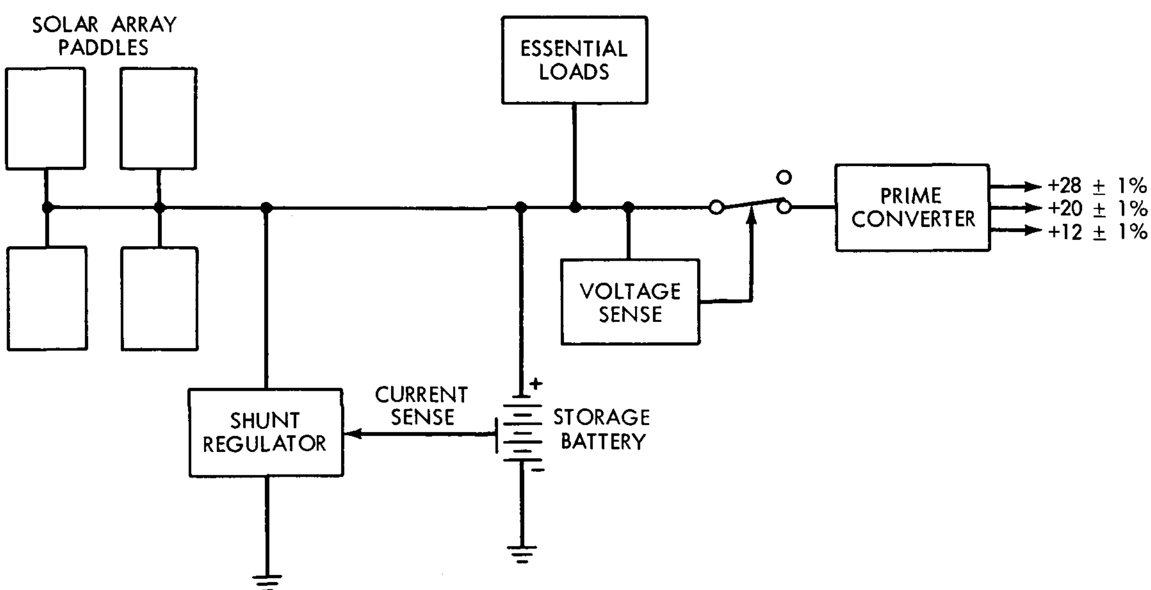


Figure 3. AIMP Power System, Block Diagram

The power profile shown in Figure 4 is a relatively flat profile. The major peak load is caused by operation of the magnetometer experiment flipper. This 39-watt peak occurs for a 10-minute period every 12 hours.

The orbit time was expected to be about 10 hours with about a 10 percent shadow time. However, orbit times up to 70 hours with down to zero percent

Table 1
AIMP Power System Weight Summary

Component	Weight (lb)
Solar array (4)	24.6
Battery	10.9
Prime converter	4.0
Solar array regulator	0.40

shadow time were acceptable, the objective being to achieve the lunar orbit. In any case the power system design was to be capable of handling the longer orbit times.

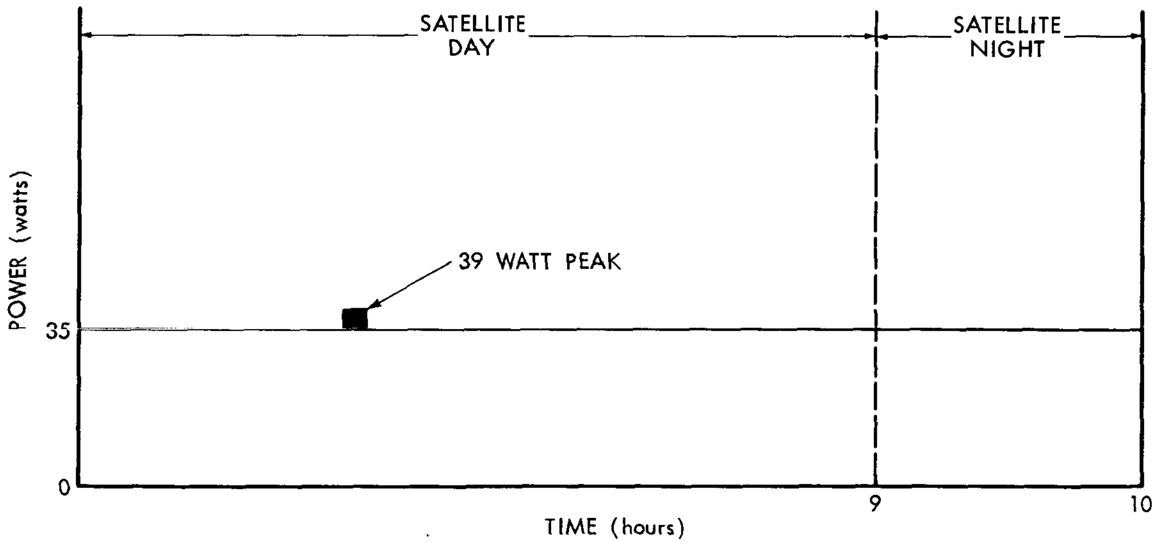
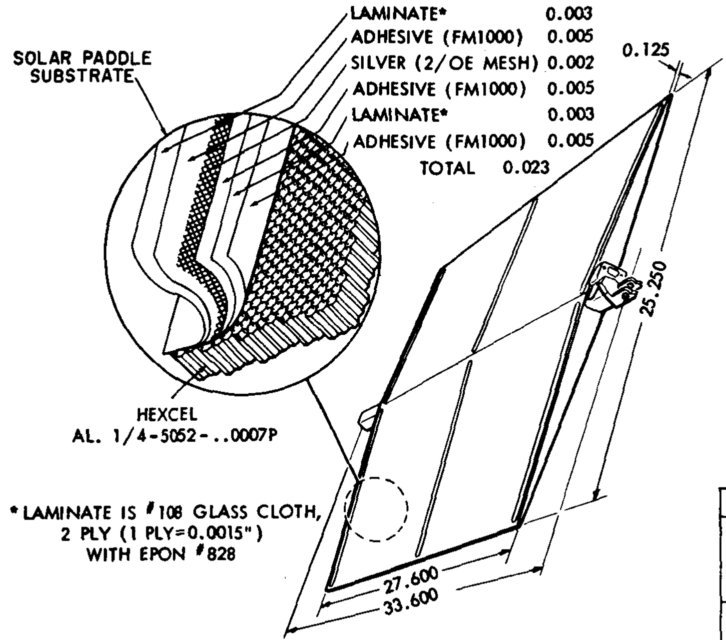
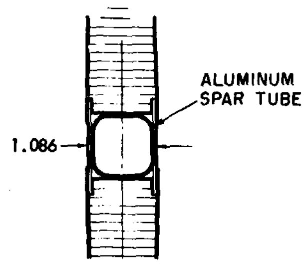


Figure 4. AIMP Power Profile

Solar Array

The solar array consists of four identical solar paddles mounted so that, in the orbital configuration, the spars extend radially outward from the spacecraft (Figure 2). The paddles are positioned around the spacecraft 90 degrees apart from each other and, to provide power at any arbitrary sunline to spin-axis angle, are pitched to an angle of 25 degrees between the plane of the paddle and the plane of the spar-spin axis.





WEIGHT BREAKDOWN

PADDLE STRUCTURE		2.72 LB.
1920 N ON P SILICON SOLAR CELLS, 2 X 2 CM.	30g / 100 CELLS	1.27 LB.
1920 COVERSLIDES CORNING 7940 FUSED SILICA, 0.006 THICK	15g / 100 CELLS	0.64 LB.
SILICONE ADHESIVE	15g / 100 CELLS	0.63 LB.
	TOTAL	5.26 LB.

Figure 5. AIMP Solar Paddle Construction

Each solar paddle is basically a rectangle, 27.60 inches long and 25.25 inches wide. Figure 5 shows construction of the solar paddle. A 1-square-inch aluminum spar tube runs the length of the paddle at the centerline providing a means for paddle attachment at the inboard end and a means for tiedown during launch at the outboard end. The solar paddle substrate is constructed of a 0.7-mil aluminum honeycomb core upon which is mounted a skin consisting of a double layer of 1.5 mil epoxy-impregnated Fiberglas and a silver, expanded-metal conductor 0.002 thick. The expanded metal, which is mounted directly beneath the solar-cell modules, provides a path for a sheet current return which, by retracing the current in the solar cells, cancels the stray magnetic field generated by that current. The complete sandwich unit eliminates the need for a separate substrate skin while still providing the structural strength necessary for launch conditions. For thermal stability, silicon adhesives are used to bond the solar cells and their glass shields, the solar cells bonded to the substrate with RTV-40 and the shields bonded to the solar cells with RTV-602.

Solar energy is converted to electrical energy by means of a total of 7680 2 cm by 2 cm solar cells divided equally among the four paddles, or 960 cells on each paddle face. The cells on each paddle are further divided into four groups, which are each isolated electrically from the others by a parallel pair of diodes. Within each group the cells are multiply interconnected with a gold-plated molybedenum bus bar into a series-parallel arrangement of 48 cells in series and 5 cells in parallel to provide redundancy and reliability.

Shadow-diffused, n-p gridded silicon solar cells are used to provide high conversion efficiency (10 percent minimum) under air mass zero solar illumination and stability in the space radiation environment. In addition, a lightweight assembly is achieved by the use of cells which are only 12.5 mils thick and which incorporate solderless titanium-silver contacts.

Each solar cell is protected by a 6-mil thick fused silica cover glass with a 410-millimicron cutoff ultraviolet filter on the cell side and an antireflecting coating on the exposed side. This arrangement provides for maximum conversion efficiency by maintaining the light-transmission characteristics of the cover glass-adhesive combination in an ultraviolet and hard-particle radiation environment, by protecting the solar cells against micrometeorite damage, and by reducing the cell operating temperature.

The power-producing capability of each solar paddle face before degradation is 42.8 watts at 19.8 volts if operated at 40°C* and oriented for maximum illumination toward the sun. At the end of 1 year, the maximum degradation anticipated is 15 percent. Since the power capability of the complete solar array varies with aspect, time, voltage, and temperature, complete details cannot be given here. However, Table 2 lists capabilities under various conditions of interest.

Table 2
AIMP Solar Array Power Output (watts)

	Normal Operation (19.6 v)		Shadow Emergence (15.5 v)	
	Avg. Power at Min. Aspect	Min. Instantaneous Pwr.	Avg. Power at Min. Aspect	Min. Instantaneous Pwr.
Initial	66.1	54.4	57.9	47.6
End of life	56.2	46.3	49.2	40.5

*A thermistor package is mounted within the spar of each solar paddle to monitor temperatures during prelaunch testing and flight.

Battery

The storage battery is a sealed nonmagnetic silver-cadmium battery rated at 11 ampere hours. It is composed of 13 cells connected in series providing a battery discharge voltage, when fully charged, of 15 volts. The maximum safe charging voltage is 19.6 volts. Continuous charging at 19.6 volts, after the battery is fully charged, is detrimental to the cells; therefore, the regulator is designed to reduce the charge to 18.3 volts upon full charge of the battery (full charge is indicated when the charge current falls below 100 milliamperes at the 19.6-volt level).

Power Control and Distribution

Solar Array Regulator—As previously mentioned, the solar array regulator functions to prevent excessive voltage generated by the solar cells from damaging the spacecraft and its battery. The regulator operates in either of two modes depending upon the state of charge of the battery. A current-monitor sensor in the battery determines the state of charge. When the battery current is in excess of 100 ma, an appropriate signal sets the regulative level at 19.6 volts. When the battery current diminishes below 100 ma the regulation is set at 18.3 volts. This lower level of regulation is used to eliminate the possibility of cell unbalance, which is a major problem with the overcharging of silver-cadmium batteries.

Undervoltage and Load Switching—Discharge below a 12-volt level is also detrimental to the battery; accordingly, an undervoltage sensing circuit detects this condition and operates a relay removing all nonessential loads from the battery. With these loads removed the full current from the array is used to charge the battery. When the battery voltage rises above 17.8 volts, the voltage sensing circuits reconnect the loads thereby reducing the charge current and returning the system to normal operation.

Prime Converter—Power from the solar array-battery bus is regulated, converted, and supplied to the spacecraft loads at three voltage levels by the prime converter. This device consists of three basic sections: a preregulator, a dc-dc converter, and dissipative regulators. The preregulator is used to accommodate the large operating voltage swing of the solar array-battery bus. The output voltage is about 11.7 volts, which is less than the 12-volt minimum input voltage. This power is then fed into the dc-dc converter which provides three separate isolated output voltages. Each voltage is then regulated to within 1 percent by independent dissipative regulators. The regulators, all similar in design, have inherent self-protection for short circuits. The regulated output voltages are +28 volts, +20 volts, and +12 volts.

Telemetry Sensors—Telemetry sensors monitor in-flight operation of the power system. The following functions are sampled: battery voltage, battery current, solar-array voltage, solar-array current, spacecraft current, 28-volt bus, 12-volt bus, solar array temperature, battery temperature, and prime converter temperature. Owing to the method of stabilization, energy balance is not an operational problem since the undervoltage detectors provide an automatic cutoff for the batteries.

Performance

Explorer XXXIII, the first AIMP, was launched on July 1, 1966. Originally intended for orbit around the moon, a slightly higher than planned thrust from the launch rocket caused the spacecraft to go into the highest earth elliptical orbit ever attained. This orbit carries the spacecraft out a distance of more than 270,000 miles from the earth's surface. The orbit period varies from 12 days to 30 days. During the first year only five periods of eclipse will occur. The maximum eclipse duration will be 50 minutes.

To date, scientific results from the spacecraft have proved conclusively that the earth's magnetosphere extends beyond the moon. A few days after launch, the satellite also recorded for the first time a shock front moving from the sun after a solar flare event.

The power system performed excellently for the first 5 months. The unexpected orbit period was not a problem because of the undervoltage protection feature. All regulating and charge-control circuits performed as expected. During the early stages of flight the temperature on the top of the spacecraft was higher than normal. This temperature eventually caused the battery temperature to rise above the design limits. In December the battery string failed, causing the spacecraft to lose power during shadow periods. Operation during satellite day was not affected by this failure. Investigation of the cause of the temperature rise is underway.

NIMBUS II

The Nimbus Program is a major research and development effort in the use of satellite technology for meteorological purposes. It features a series of earth-oriented spacecraft carrying a variety of experiments designed to observe and measure the earth's atmosphere and to transmit rapidly the collected data. The Nimbus spacecraft (Figure 6) are launched in a sun-synchronous near-polar circular orbit. This orbit permits complete global television picture coverage of the daytime cloudcover every 24 hours. Measurements of the infrared and reflected radiation and of the earth's heat balance are also made every 24 hours.

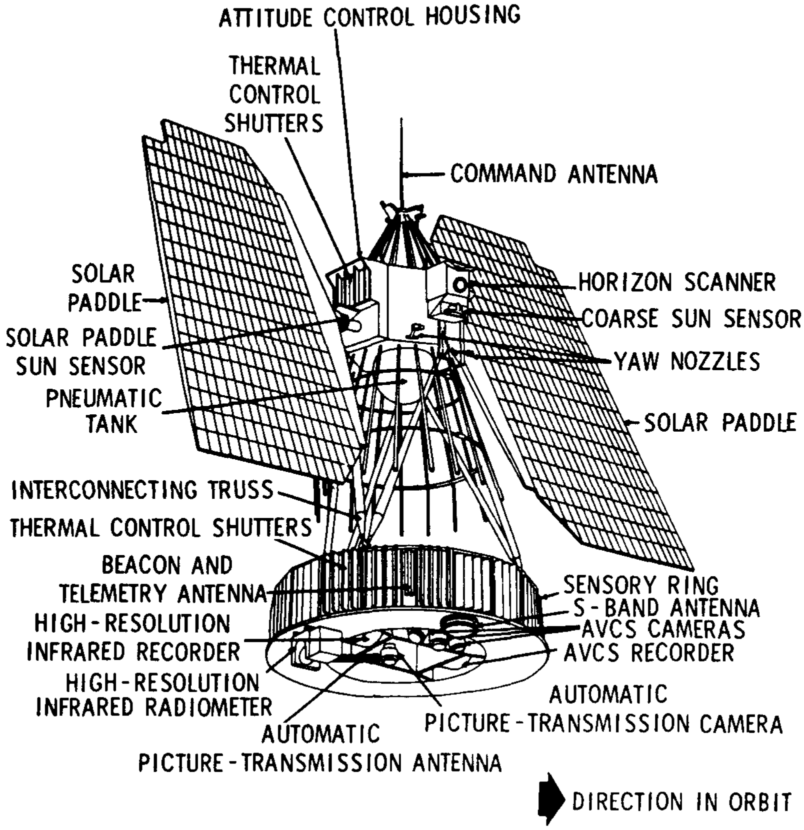


Figure 6. Nimbus Spacecraft

A second feature of the orbit is that a single axis of rotation may be used to provide a completely oriented solar array. This feature results in a large reduction of the array size and weight.

Power System Description

The power system for Nimbus II is designed to provide no less than 160 watts of continuous spacecraft power for a minimum lifetime of 6 months. Because an active control system is used to orient the spacecraft and the solar array, a minimum power level of 100 watts is required for spacecraft survival.

Figure 7 shows the power demand profile for a typical Nimbus spacecraft. Several features of this profile are noteworthy: First, as mentioned, the spacecraft has a minimum load which the input power must meet or the spacecraft will be lost. Second, the experiment or sensor load falls naturally into two sections, an earth-day load and an earth-night load. The earth-day load consists of those sensors which are used to view the illuminated portions of the earth; these are turned on automatically just after the spacecraft begins to view the sunlit portion

NOTE:
CROSS-HATCHED AREA REPRESENTS AVERAGE INTERROGATION LOAD. DURATION OF THIS LOAD IS VARIABLE. FOR A GIVEN 24-HOUR PERIOD, THERE ARE 14 ORBITS AT 500 NAUTICAL MILES. OF THESE, 8 ORBITS HAVE 6.8-MINUTE INTERROGATION PERIODS; 1 ORBIT HAS AN 8-MINUTE INTERROGATION PERIOD WITH RESPECT TO S-BAND TRANSMISSION. (PREDOMINANTLY 4 CONSECUTIVE ORBITS CANNOT BE INTERROGATED).

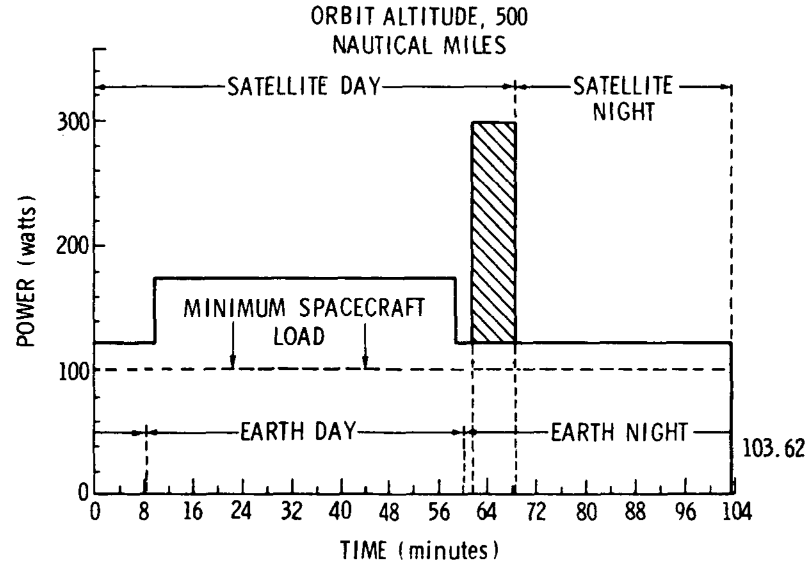


Figure 7. Nimbus II Power Demand Profile

of the earth, and are turned off just before the spacecraft begins to view the dark side of the earth. Conversely, the nightload sensors are intended to view the dark portion of the earth; these are also automatically turned on and off as the spacecraft moves into proper position.

To attain complete earth coverage, television and infrared data are stored on magnetic tape and then transmitted to earth once each orbit. To take 100 minutes of data, and transmit these data to earth during a 5-minute pass over a ground station, requires a high-rate data-transmission system. Such a system uses a large amount of power, hence, the large power spike shown in the demand profile. Note that the time appearance of this spike is a function of many variables such as season of launch, interrogating ground-station location, and operational plan. The pulse can occur entirely in the sunlight, partially in sunlight, partially in dark, or entirely in the dark. The power system must be designed to handle all these cases.

The stated requirements are fulfilled by two major groupings of hardware: two solar-cell platforms attached to the control subsystem housing by platform driveshafts, and sensory ring hardware consisting of seven parallel-connected battery modules and a control or electronics module. Figure 8 is a functional block diagram of the Nimbus II power system. The sensory ring, through a compartmented design and simple interface connections, provides a housing and independent thermal-control capability for each module. Table 3 shows physical dimensions and weights for each power-supply assembly and for the total system.

Table 3
Size and Weight of Nimbus II Power
System Assemblies

Assembly	Dimensions				Weight (lb)
	Width (in.)	Height (in.)	Depth (in.)	Volume (cu. in.)	
Single solar cell platform, including transition section	46.75	96			38.5
Battery module	6	8	6-1/2	312	15.2
Electronics module	6	4	13	312	6.8

Total power system weight is 190 lbs.

Solar Array

Each solar-cell platform consists of 5472 2 cm by 2 cm phosphorus doped silicon n-p cells, a mounting structure, a transition section, a latching assembly, a drive motor with an associated gear-reduction unit, and a control-shaft clamp. The solar cells, mounted on one side of the aluminum honeycomb platforms and maintained continuously incident to the earth-sun line, can intercept a maximum of solar energy during the sunlight period. Assuming a solar-cell module efficiency of 10.1 percent at air mass zero (25°C), the environment in space, and the power supply operating characteristics, the array can supply approximately 460 watts at the design operating point at a temperature of 55°C at the beginning of life.

The solar cells are in a series-parallel configuration. Each cell contains a 6-mil fused silicon cover with a blue-red filter coating. The covers are held in place with Furane 15-E cement.

The basic building block is a flat-mounted solar-cell module. The module consists of 10 cells connected in parallel. Ninety-eight of the modules are connected in series to form a board, and each platform contains five of these

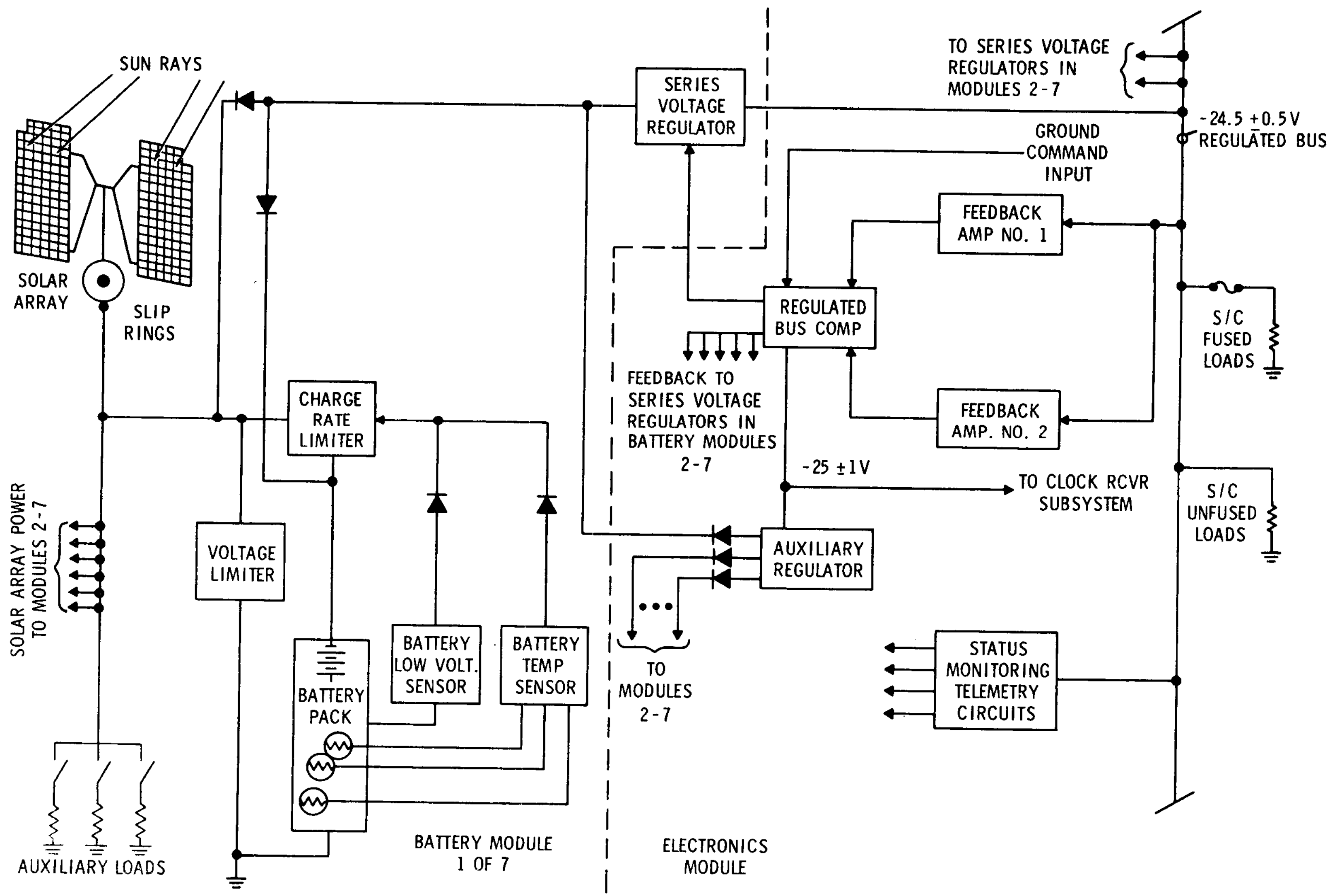


Figure 8. Nimbus II Power System, Simplified Block Diagram

boards. Each platform also contains a sixth board, called the long board, made of 6-cell and 3-cell modules to make maximum use of available surface area. Each board is connected to the solar array bus through isolation diodes. Figure 9 is a diagram of the board configuration on each paddle.

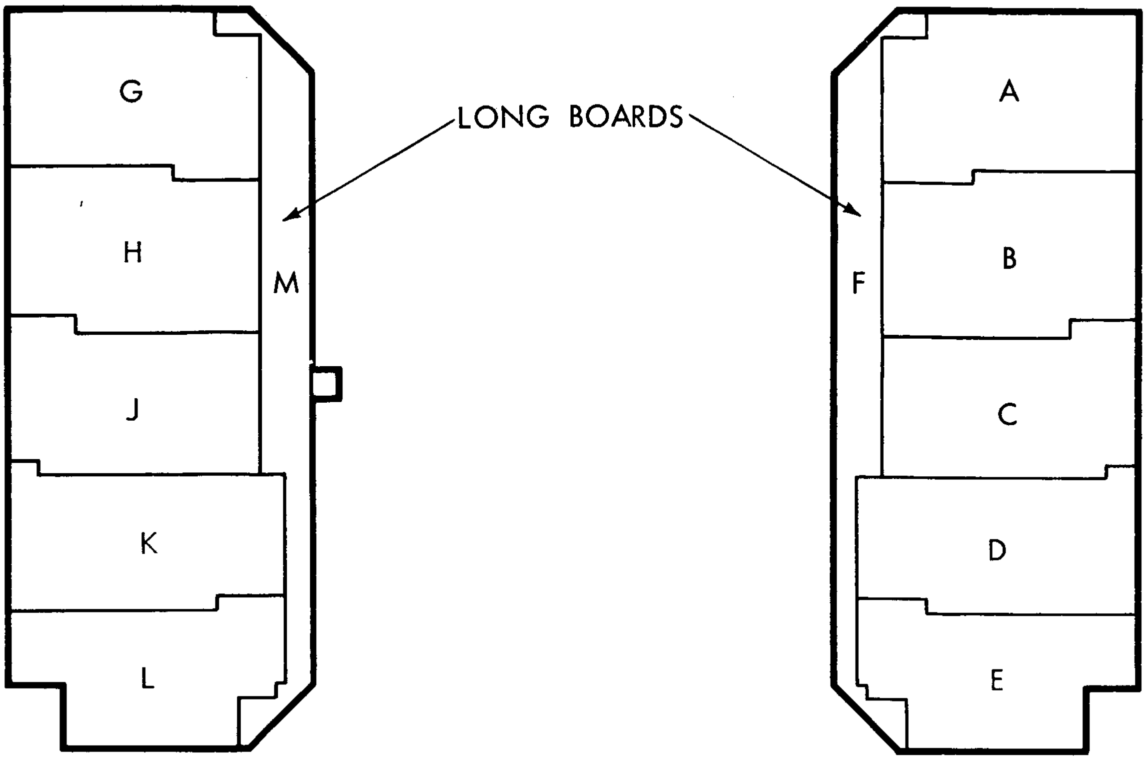


Figure 9. Nimbus Paddle Configuration

The Nimbus II solar array output (worst case conditions) are as follows:

- Initial: 449 watts at 40°C, 39.69 volts
- End of life: 387 watts at 40°C, 39.45 volts

Battery

The Nimbus system concept includes storage of energy in multiple parallel-connected battery packs to provide fail-safe redundancy. A battery module contains a single battery pack composed of 23 series-connected nickel-cadmium cells. Nominal capacity of each cell is 4.5 ampere-hours. Depth-of-discharge is limited to a maximum of 20 percent during normal system operation. If a battery pack fails in a semi-to-full short mode, the failed unit automatically adjusts to accept only a trickle charge during satellite day, and automatically isolates itself from the remaining units during satellite night, so that normal system operation may continue. Six battery packs were sufficient to sustain full spacecraft system capability for the Nimbus II mission.

Inherent in the multiple battery-module concept is individual charge control for each battery pack. A charge regulator limits the maximum battery-charge current to 1.5 amperes. Two measures employed for battery protection can individually reduce the charging current to a nominal trickle rate of 300 ma. Three sensors strategically placed in the pack monitor the pack temperature. If any one of the sensors reaches 65°C, the pack is switched to the trickle rate. Battery voltage is continuously monitored and, at a pack voltage of 21 volts or below, a trickle rate will be maintained. This guarantees maximum use of available system-recharge current should a battery pack fail catastrophically. The Nimbus battery modules also contain solar-array power limiters and status-monitoring telemetry circuits. A total system limiter consists of seven parallel circuits to provide redundancy and maximum protection against catastrophic failures. The limiter prevents voltage excursions of the array beyond 40 volts to protect the various regulation circuits from over-dissipation caused by excessive array voltage. Circuits whose status is monitored to ascertain system performance include battery temperature, battery voltage, and battery-charge and battery-discharge current. The output of each status-monitoring point is suitably conditioned for input to the telemetry subsystem.

Power Control and Distribution

The electronics module performs a centralized power-control and distribution function for the subsystem. Each spacecraft subsystem receives power from a common bus at a -24.5 volt ±2 percent level. Exothermic pyrofuses are provided for each noncritical load. Power control is accomplished by a dissipative regulation system. Each of the battery modules contains a separate load-current-regulating element preadjusted to share the total load current and driven by a common load-bus-derived error signal. Individual regulator elements, fused for short-circuit protection, are monitored by telemetry to indicate overall status. An isolated connection to the input of each element from its associated battery pack and/or the solar array is made through a disconnect diode. The regulated bus voltage is sensed in the normal manner, and one of two redundant feedback amplifiers feeds back an error signal to the regulating elements. A regulated bus comparator senses the main bus voltage for low or high limit conditions; if the bus voltage exceeds the design limits, an error-switching signal initiates switchover to the redundant amplifier. A ground-commanded switchover facility is provided as an emergency backup.

An auxiliary regulator, receiving a proportionate amount of input power from each of the battery modules, provides power for the comparator and associated ground-command channels in the spacecraft clock receiver. The independent power source ensures proper comparator and clock-receiver operation should a sudden loss of the main bus occur. The electronics module also provides status-monitoring circuits, including unregulated and regulated bus voltages and currents, and the operational status of the feedback amplifiers.

Auxiliary loads are used to provide battery-charge control. With a known array current and a defined spacecraft load the amount of current required to bring the batteries back to the full charge condition is computed. The auxiliary loads are then set to bleed off the excess array current. These loads can be readjusted each orbit if necessary.

Performance

Nimbus II was successfully launched on May 15, 1966. The spacecraft achieved a near perfect orbit and initial power system operation was well within specifications. Voltage regulations and energy balance of the system have been maintained for more then 9 months of continuous operation.

Some difficulty was encountered when the calibration of the telemetry current sensors shifted; however, the errors were detected and corrected for on the ground. In orbit 1066 one of the series interconnections on board M opened causing a loss of 700 mils of array current. The loss was not catastrophic, and the spacecraft can still be powered. A paper on the flight performance of the system is being prepared.

ATS I

The Applications Technology Spacecraft program was created to advance the state of the art in spacecraft technology. The program consists of five spacecraft having one of the following orbit configurations:

- Spin-stabilized spacecraft with synchronous equatorial orbit
- Gravity-gradient stabilized spacecraft with a 6000-nautical-mile altitude orbit inclined approximately 28.5 degrees
- Gravity-gradient stabilized spacecraft in an equatorial synchronous orbit

The ATS program is a logical continuation of previous work in space communications, observations of space phenomena, meteorological observations, orbit control, spacecraft stabilization, and spacecraft-damage experiments.

In the communications field the Syncom series has shown the feasibility of communication by means of stationary satellites. Based on preliminary studies of new communications techniques, the ATS program will evaluate the capability of a single-sideband multiple-access system as well as that of a high-quality frequency-modulated system which is a refinement of the Telstar, Relay, and Syncom systems. In the meteorological field the ATS will provide an ideal platform for research since the subsatellite point remains essentially stationary. The results of this experiment, combined with the knowledge gained from the experiments on TIROS and Nimbus, will provide the first step in evaluating meteorological experiments having synchronous orbits. In the environmental field the ATS spacecraft will carry groups of scientific experiments which will use for the first time a platform stationary in both geocentric and geomagnetic fields. Data from these experiments will provide a scientific tiepoint in the earth's environment with which data from previous scientific series such as Explorer, Pioneer, and IMP may be correlated.

The first spacecraft in the series was the spin-stabilized spacecraft in a synchronous orbit. The spacecraft is illustrated in Figure 10, which locates the various experiments. The experiments included a phased-array antenna, environmental measurements, VHF repeater, ion engine, spin-scan cloud camera, and a nutation experiment.

Power System Description

A solar array and rechargeable nickel-cadmium batteries provide electric power for all three versions of the ATS spacecraft. The n-p silicon solar cells provide the primary source of electric energy; the batteries provide electric power during transient loads and solar eclipse. Sufficient solar-cell capacity has been provided to account for system degradation due to particular radiation in the orbital environments, boom shadowing, and sun angles up to 23 degrees from normal.

The electrical power systems of the three spacecraft systems are schematically identical, with some physical difference in the number of solar cells used to charge the batteries. Figure 11 is a block diagram of the power system. Each main solar array directly powers an unregulated spacecraft bus whose voltage range is maintained between -32.5 and -24.5 volts. The upper limit is controlled by the bus voltage limiters and the lower by the battery-discharge control circuit. Each battery is charged directly by a small solar-cell battery charge array, which provides the required charge voltage and which also acts as a battery-charge current limiter. The bus relay automatically unparallels the two busses whenever the bus voltage falls below a nominal value of -22 volts. The busses can be unparalleled or reparalleled on command.

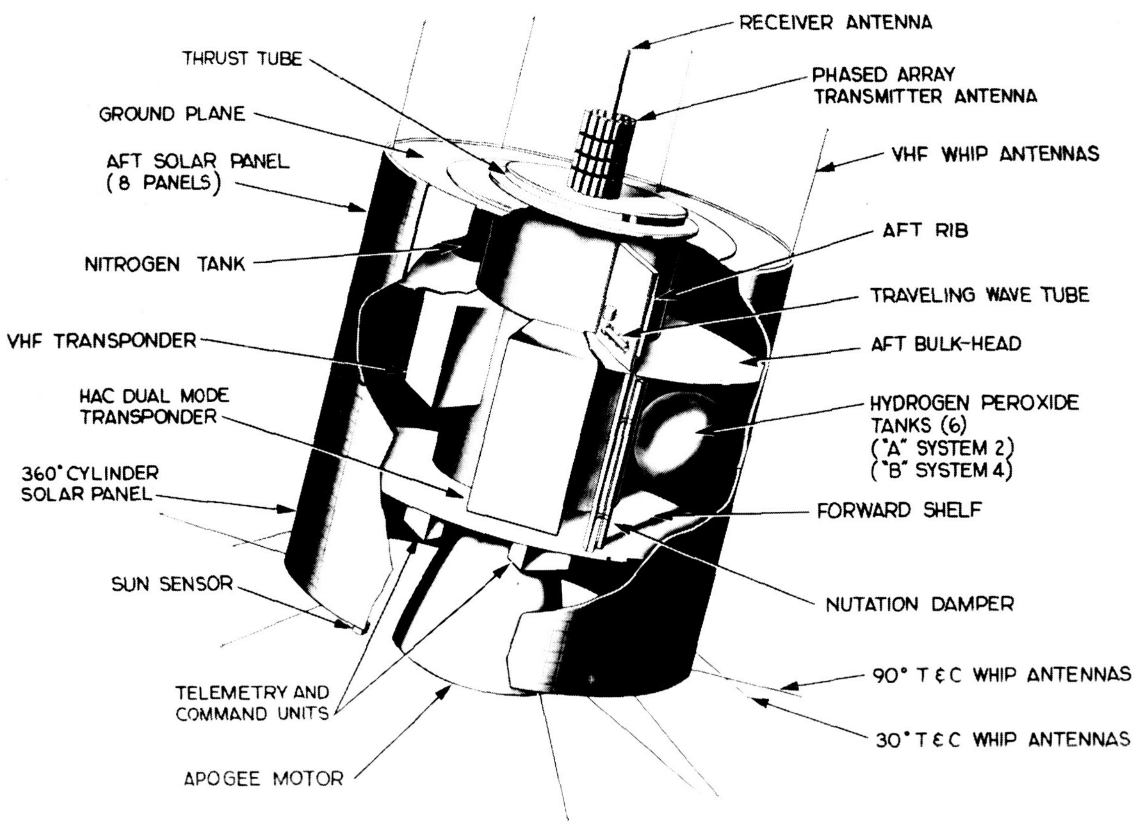


Figure 10. ATS Synchronous Altitude Spin-Stabilized Spacecraft

There is no normal operating power profile for the ATS-I owing to the large number of experiments. The minimum load for the spacecraft is 10 watts. With all loads turned on the power demand is 320 watts. The average load will depend upon ground programming of the experiments and will probably average about 160 watts.

Table 4 lists the weights of the various components of the ATS-I power system.

Table 4
ATS-I Power System Weight Summary

Component	Weight (lb)
Solar array	61.4
Batteries (2)	36.0
Discharge controllers (2)	7.58
Voltage limiters (2)	2.74

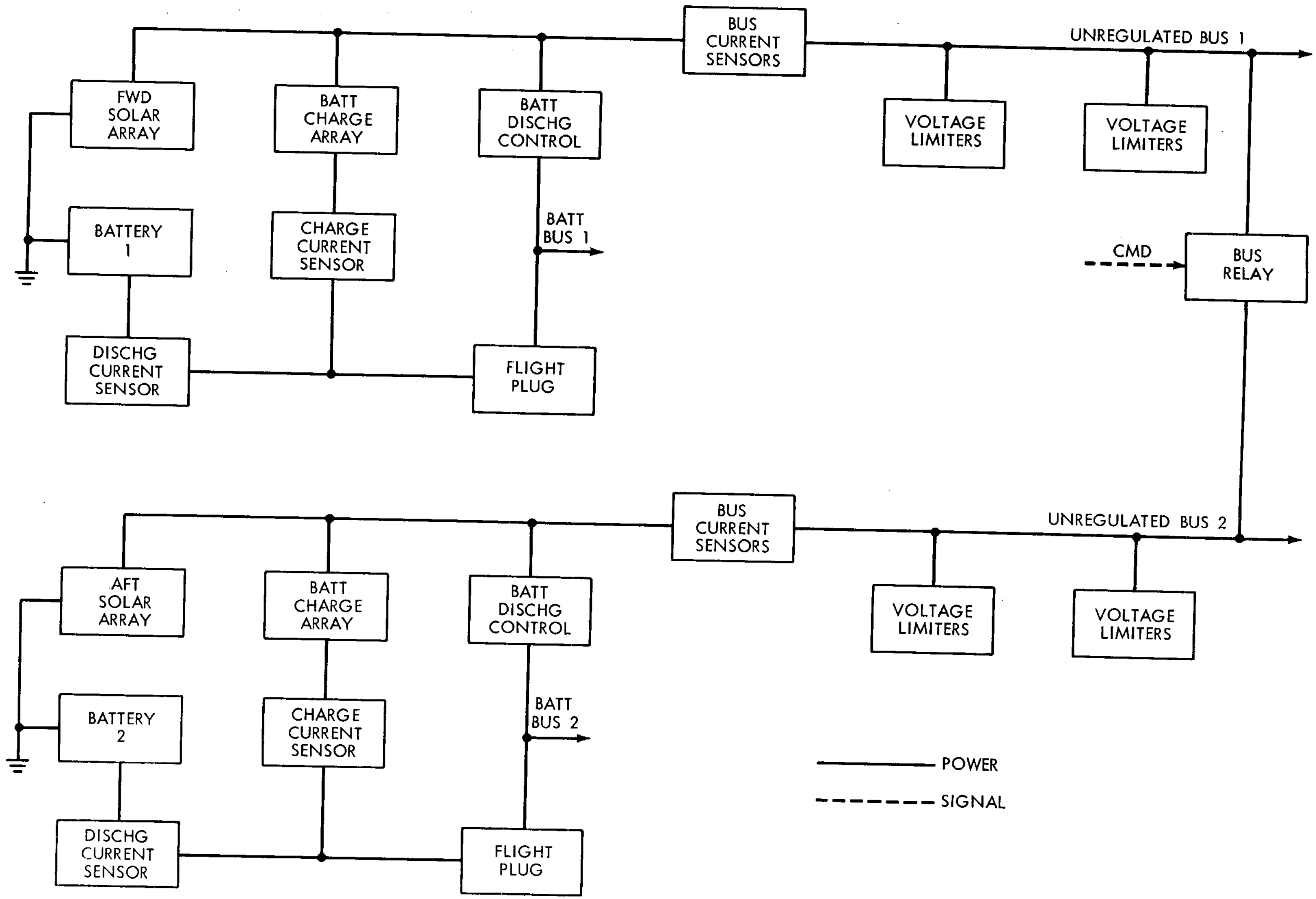


Figure 11. ATS Electrical Power System, Block Diagram

Solar Array

The basic cell for the ATS-I spacecraft is a 1 cm by 2 cm silicon n-p solar cell. The base resistivity of these cells is 10 ohm-centimeter, and each cell contains a fused silica cover 30 mils thick. The cover also contains a blue filter and antireflective coatings.

The three solar arrays comprise the outer cylindrical shell of the spacecraft (Figure 12). The aft array is composed of eight individual panels each of 45 degrees arc. The panels are easily removable to permit access to the payload section of the spacecraft. The forward main array consists of 66 parallel-connected subgroups of cells. Each subgroup is made of three parallel strings of 62 cells in series running the full length of the panel. The aft segmented array contains a total of 187 strings arranged mainly in parallel groups of three but also in single and double strings where necessary to utilize all the panel area. Each subgroup of solar cells is connected to the unregulated bus through

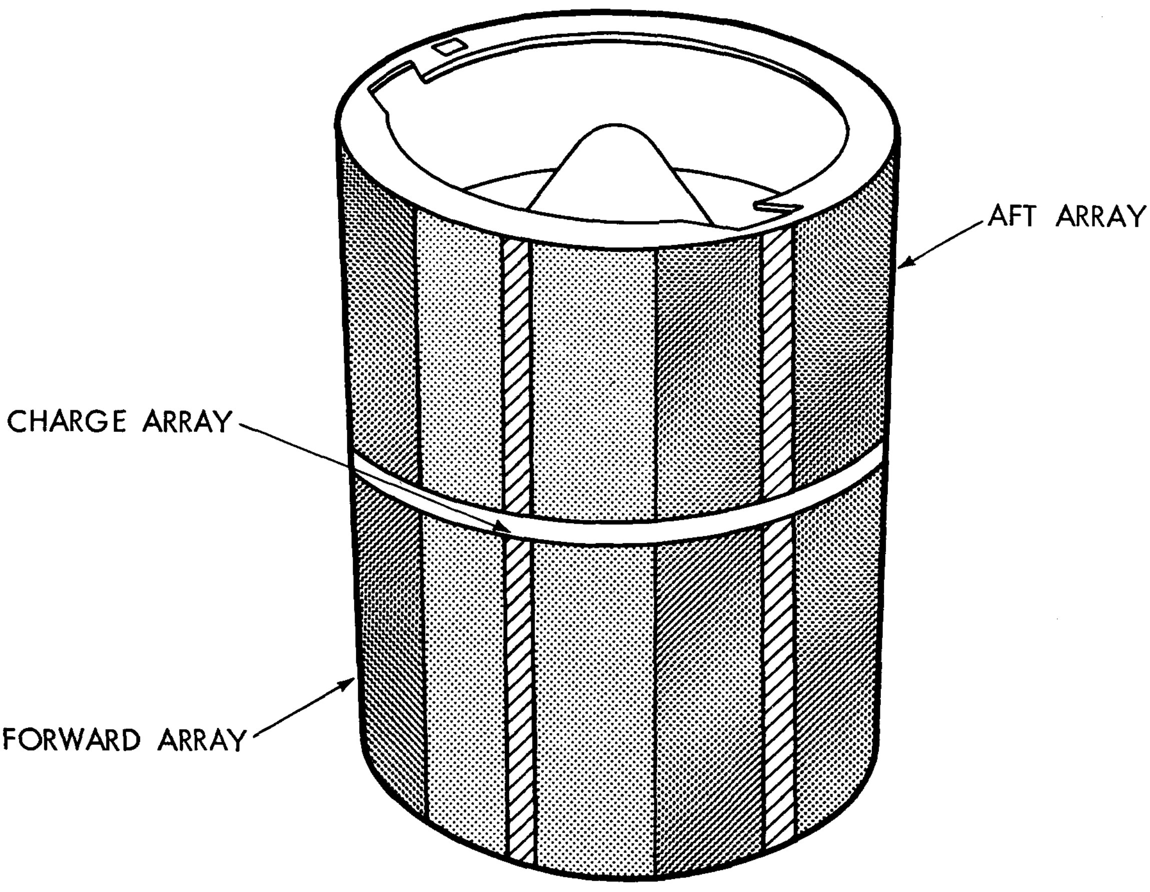


Figure 12. ATS-I Solar Array Locations

a pair of high conductance silicon diodes. Both the main and aft array have a small separate array for battery charging. Each battery charge array contains 12 strings of 15 series-connected cells. The strings are symmetrically spaced on strips of one cell width at 60-degree intervals around each solar array cylinder.

Figure 13 shows the performance curve for the solar array output.

Battery

Each unregulated spacecraft bus is powered during high transient and eclipse load operation by a nickel-cadmium storage battery. Each battery consists of 22 series-connected 6-ampere-hour cells. Each cell contains a hermetic seal.

The orbital period is 24 hours, and the maximum shadow time is 1.15 hours. Thus, the maximum discharge current capability during the longest eclipse period (to allow for full recharge on each orbit) is 2.35 amperes per battery. This results in a battery depth of discharge of 45 percent for the longest eclipse. The semiannual eclipse season at synchronous altitude is only 45 days; therefore, the cycle life requirements over the 3-year expected life are only 270 cycles.

Power Control and Distribution

Battery-Discharge Control—The discharge regime of the spacecraft batteries is controlled by integral battery-discharge controls. These controls allow the batteries to discharge only when their associated spacecraft busses approach -24.5 volts. The use of this discharge control allows for the complete use of the maximum power available from the solar array without discharging the spacecraft batteries. The spacecraft batteries will furnish only the additional current requirements that the main array cannot furnish.

Current Sensors—Eight current sensors provide general housekeeping data for the performance of the power system aboard the spacecraft. The power of each solar array bus is carried by two bus wires; thus, two sensors are used per bus to determine the total solar array current. The output of the two bus sensors are added together in the spacecraft before entering the encoder. In addition, one sensor is used for battery charging information and one for battery discharging information. Separate sensors determine the battery currents, instead of a single biased sensor, to provide more accurate information on battery charging, since the battery charge and discharge requirements vary from one another by an order of magnitude. The bus current sensors are of the magnetic amplifier type.

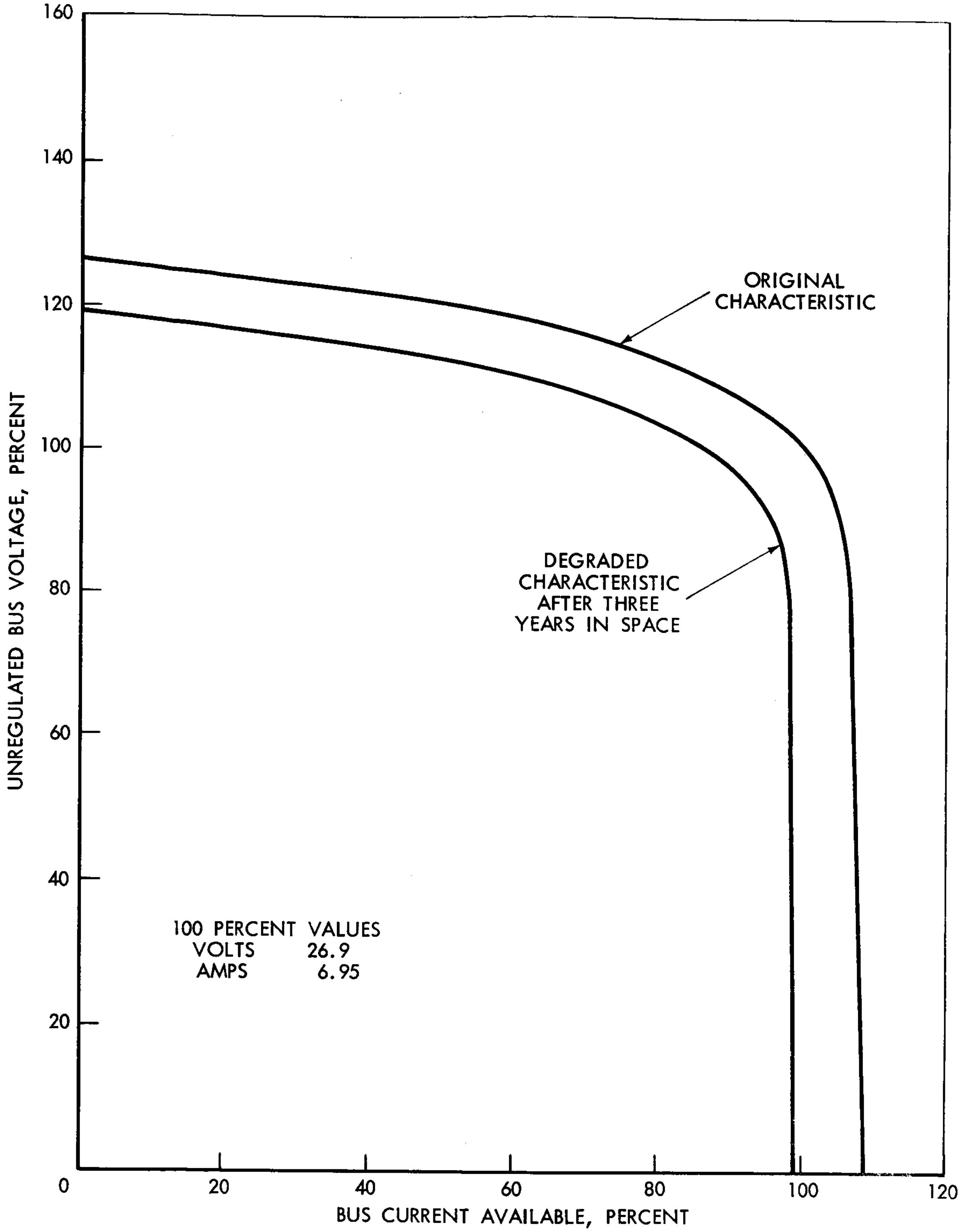


Figure 13. ATS-1 Solar Array Output Performance Curve

Solar Array Voltage Limiter Units—Voltage limiter units, composed of essentially the same components as used in the battery-discharge control, have been added to the spacecraft to limit the maximum voltage that the unregulated bus will see during any transient period of operations, such as emerging from the eclipse. The spacecraft contains eight units, with two units packed together physically to provide thermal inputs to the spacecraft in key locations.

Bus Paralleling Relay—A relay is included in the spacecraft to provide greater flexibility of payload and electronic subsystem operation. The potential combinations of payload operation could be asymmetric and, without the capability of tying the busses together electrically, operational restrictions would occur. The bus relay is maintained in an open position during launch. After injection into orbit, the relay can be closed and latched upon ground command. However, should a bus fault develop, the relay will automatically open and will not reclose except by ground command.

Performance

The ATS-I spacecraft was successfully launched on December 6, 1966. Spacecraft performance has been excellent, and all systems are operating properly. The power system has operated within its design limits as the spacecraft loads have been programmed up to the point of battery discharge. Tests are planned to increase the load to force discharge of the battery for short periods of time to check the performance of that portion of the system.

The normal eclipse period will start about the end of February 1967 and will continue for 44 days, with a maximum eclipse time of 72 minutes occurring on March 22.

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